[redacted]

REC'D APR 23 1952

[redacted]

Copy 1
RM E52D02

NACA RM E52D02

RESEARCH MEMORANDUM

CLASSIFICATION CHANGED
UNCLASSIFIED

To NASA TD70-688
By Authority of NASA 70-688 Date 11/25/70

ANALYSIS OF A LIQUID-METAL TURBINE-PROPELLER CYCLE FOR PROPULSION OF LOW-SPEED NUCLEAR-POWERED AIRCRAFT

By William W. Wachtl and Frank E. Rom

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

Declassified by authority of NASA
Classification Change Notices No. 212
Dated **

[redacted]

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

MAY 27 1952

14

[redacted]

 [redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

ANALYSIS OF A LIQUID-METAL TURBINE-PROPELLER CYCLE FOR PROPULSION OF LOW-SPEED NUCLEAR-POWERED AIRCRAFT

By William W. Wachtl and Frank E. Rom

2480

SUMMARY

An analysis of a nuclear-powered liquid-metal turbine-propeller cycle is presented for a range of engine operating conditions at a flight Mach number of 0.5 and an altitude of 30,000 feet.

The thrust per engine-plus-heat-exchanger weight is optimized for minimum airplane gross weight for compressor pressure ratio, heat-exchanger inlet Mach number, and turbine-inlet temperature for a range of liquid-metal-to-air heat-exchanger effective wall temperatures.

Airplane gross weight and reactor heat release is presented for typical values of airplane lift-drag ratio, structure- to gross-weight ratio, and sum of reactor, shield, payload, and auxiliary weight.

INTRODUCTION

Analyses are being made at the NACA Lewis laboratory to determine the characteristics of various aircraft propulsion systems utilizing nuclear energy. Design point studies have been made of various cycles at altitudes of 30,000 and 50,000 feet and flight Mach numbers of 0.9 and 1.5. The direct-air turbojet cycle is presented in references 1 to 3. A preliminary comparison of the direct-air, helium, and liquid-metal turbojet cycles is made in reference 3. A detailed design point study of the liquid-metal turbojet at various altitudes and flight Mach numbers is made in reference 4.

The intermediate subsonic speed range is considered and the liquid-metal turbine-propeller cycle discussed in this report. The flight Mach number is 0.5 and the altitude is 30,000 feet. The assumptions and methods of calculation are, in general, those used in reference 4 to calculate the turbojet cycle. In this report, compressor pressure ratio, heat-exchanger air-inlet Mach number, and turbine-inlet temperature were optimized for maximum engine net thrust per engine-plus-heat-exchanger weight (minimum airplane gross weight) for a range of heat-exchanger effective wall temperature. [redacted]

[redacted]

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
c_p	specific heat at constant pressure, Btu/(lb)(°R)
D	drag, lb
d	hydraulic diameter of tubes, ft
F	thrust, lb
f	free flow ratio, flow area divided by frontal area
g	acceleration due to gravity, 32.2 ft/sec^2
HP	shaft horsepower
h	heat-transfer coefficient, Btu/(sec)(sq ft)(°R)
k	thermal conductivity, Btu/(sec)(sq ft)(°R)/(ft)
L	lift, lb
l	tube length, ft
M	Mach number
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
Q	reactor heat-release rate, Btu/sec
S	surface area, sq ft
T	total temperature, °R
U	over-all heat-transfer coefficient, Btu/(sec)(sq ft)(°R)
V	velocity, ft/sec
W	weight, lb
w	weight flow, lb/sec
η	efficiency
ρ	density, lb/cu ft

Subscripts:

a	air flow
c	compressor
f	frontal
G	gearbox
g	gross
j	jet
K	shield plus reactor plus payload plus auxiliary
l	liquid metal
m	reactor maximum
n	net
p	propeller
r	reactor
s	structure
T	engine plus heat exchanger
t	turbine
w	heat-exchanger effective wall
x	exchanger
∞	small stage
0	free stream
1	compressor inlet
2	compressor outlet
2'	heat-exchanger inlet
3	turbine inlet
4	turbine outlet

[redacted]

DESCRIPTION OF CYCLE

A schematic diagram of the liquid-metal turbine-propeller cycle is shown in figure 1. A nuclear reactor is used as the heat source, and a heat exchanger replaces the standard engine combustion chamber. In general, a binary system is considered incorporating a closed liquid-metal cycle and an open air cycle; however, the effect of adding an intermediate heat exchanger is also investigated.

The liquid-metal coolant (lithium) is pumped through the reactor where it is heated by contact with the walls of the reactor flow passages. From the reactor the liquid-metal flows through the heat exchanger where heat is transferred to the air; it is then directed back to the reactor, thus completing its cycle.

The engine air enters the diffuser, is compressed by the compressor, and then passes through the heat exchanger taking heat from the liquid-metal. The hot compressed air expands through the turbine, which extracts energy to drive the compressor and the propeller. Most of the engine thrust is provided by the propeller; however, the energy which remains in the hot gas is converted to additional jet thrust by expanding through the exhaust nozzle.

ASSUMPTIONS AND METHODS

The assumptions made for the turbine-propeller cycle along with the methods of analysis are given in the following paragraphs. The methods used in the analysis for the binary cycle are, in general, those given in reference 4; consequently, only a brief mention of them is made where additions or variations were necessary. The method used to calculate the performance of the intermediate heat exchanger for the ternary cycle is shown in the appendix.

Engine

Engine efficiencies. - The efficiencies assumed for the engine components are as follows:

Propeller efficiency, η_p	0.90
Diffuser total-pressure ratio, P_1/P_0	0.98
Compressor small-stage efficiency, $\eta_{c,\infty}$	0.88
Turbine small-stage efficiency, $\eta_{t,\infty}$	0.88
Exhaust-nozzle velocity coefficient (full expansion)	0.97
Gear efficiency at design condition, η_G	0.94

[redacted]

The gearbox was charged with a constant horsepower loss based on a gearbox designed to transmit the necessary take-off horsepower developed by the engine. This assumption results in the apparently low value of gearbox efficiency listed. The efficiency at sea-level take-off was assumed to be 0.98.

Engine thrust. - In calculating the engine thrust per pound of air
2480 per second from the assumptions previously listed, the heat-exchanger pressure drop was obtained from charts presented in reference 4 and based on material from reference 5. In addition, the tail-pipe pressure ratio P_4/p_0 was assumed to be equal to the ram pressure ratio P_1/p_0. According to reference 6, this assumption gives approximately the optimum division of power between the jet and the propeller. The turbine work in excess of that required to drive the compressor is delivered through the gearbox to the propeller. The total engine thrust per pound of air per second was calculated by adding the exhaust jet thrust per pound of air per second to the propeller thrust per pound of air per second and subtracting the inlet momentum per pound of air per second:

$$\frac{F_n}{w_a} = \frac{HP}{w_a}\frac{550}{V}\eta_p\eta_G + \frac{F_j}{w_a} - \frac{V}{g} \tag{1}$$

Engine weight. - The engine weight per pound of air flow was calculated from the following equations based on the component weights of current axial-flow engines. The weights were assumed proportional to the number of stages and the frontal area of the components. The compressor weight including shaft and frame per pound of air per second is assumed to be

$$\frac{W_c}{w_a} = 465\,\frac{\sqrt{T_1}}{P_1}\ln\left(\frac{P_2}{P_1}\right) \tag{2}$$

The turbine weight including shaft and frame per pound of air per second is assumed to be

$$\frac{W_t}{w_a} = 1313\,\frac{\sqrt{T_3}}{P_3}\ln\left(\frac{P_3}{P_4}\right) \tag{3}$$

The propeller and gearbox weight were based on a propeller and a gearbox designed to transmit the necessary take-off horsepower developed by the engine. This weight was then charged to the design point engine. Based on a per pound of air per second basis the weight is

$$\frac{W_{G+p}}{w_a} = 0.633\left(\frac{HP}{w_a}\right) \tag{4}$$

[redacted]

Heat-exchanger assumptions and weight. - The liquid-metal-to-air heat exchanger is assumed to be of the tubular counterflow type with air flowing through the tubes and lithium flowing in the spaces between the tubes. The tubes were assumed to be made of stainless steel having an internal diameter of 0.25 inch and a wall thickness of 0.01 inch. The exchanger free flow ratio $(A_a/A_f)_x$ is assumed to be 0.65.

The exchanger is assumed to have a constant effective wall temperature equal to the log mean effective wall temperature. The exchanger length-diameter ratio l/d and the total-pressure ratio P_3/P_2 can then be computed from charts in reference 4 based on material from reference 5.

The exchanger weight (including coolant and headers) per pound of air per second is obtained from the following equation:

$$\frac{W_x}{w_a} = 1.9\left(\frac{l}{d}\,\frac{A_a}{w_a}\right)_x \tag{5}$$

where the constant 1.9 is evaluated from the exchanger tube diameter, tube wall thickness, free flow factor, and lithium weight.

Engine thrust per total engine weight. - The engine performance was evaluated in this report by optimizing the thrust per engine plus heat exchanger weight F_n/W_T. This factor is determined by dividing the thrust per pound of air per second by the required weight of the engine plus heat exchanger per pound of air per second.

Engine frontal area. - The frontal area of the compressor was calculated assuming that the air flow corrected to sea-level static conditions per unit frontal area is 25 pounds per second per square foot. The heat-exchanger frontal area was calculated from the air inlet Mach number into the exchanger tubes and the exchanger free flow ratio $(A_a/A_f)_x$. The turbine outlet area was calculated assuming a hub-tip ratio of 0.6 and an air outlet Mach number of 0.70. The nozzle outlet area was determined assuming full expansion and a nozzle pressure ratio equal to the ram pressure ratio. The engine frontal area is then determined by the largest component.

Airplane gross weight. - The airplane gross weight is obtained from the following equation as given in reference 4:

$$W_g = \frac{W_K}{1 - \left(\frac{W_s}{W_g}\right) - \frac{1}{\left(\frac{L}{D}\,\frac{F_n}{W_T}\right)}} \tag{6}$$

Figure 2 has been prepared from equation (6) for a value of W_s/W_g of 0.35, so that W_g can be conveniently determined for any given values of W_K, F_n/W_T, and L/D. Airplane gross weight is determined by multiplying W_g/W_K, evaluated at the desired value of F_n/W_T and L/D, by any assigned value of W_K.

2480

Airplane drag and weight assumptions. - The airplane design point lift-drag ratio including nacelles is assumed to be 20 for the flight Mach number of 0.5. The airplane structure-gross-weight ratio is assumed to be 0.35. The symbol W_K is defined as the sum of reactor, shield assembly, payload, and auxiliary weights (pumps, piping, chemical fuel, etc). Airplane performance in terms of gross weight W_g, reactor heat release rate Q, total engine air flow w_a, and engine component frontal areas A_f (compressor, air heat exchanger, turbine exit, and jet nozzle exit) is presented in terms of a ratio to W_K. Therefore W_g, Q, w_a, and A_f can be obtained by multiplying the corresponding ratio by any assigned value of W_K. The corresponding reactor maximum wall temperatures are also presented as a function of W_K.

Reactor heat release. - The required reactor heat release was computed from W_g, L/D, F_n/w_a, and heat addition per second to each pound of air per second through the heat exchanger Q/w_a as follows:

$$Q = w_a\left(\frac{Q}{w_a}\right) = \left(\frac{W_g}{\frac{L}{D}\,\frac{F_n}{w_a}}\right)\left(\frac{Q}{w_a}\right) \tag{7}$$

No heat losses in the piping system were considered in this report.

Reactor maximum wall temperature. - The reactor maximum wall temperature is computed by making the following assumptions: (1) the difference between the reactor wall temperature and the liquid-metal temperature is constant along the reactor flow passages; (2) the heat-exchanger effective wall temperature is equal to the average of the inlet and outlet coolant temperature; (3) the coolant velocity is fixed at 15 feet per second; (4) the reactor diameter and length are 2.5 feet; (5) the reactor flow passage equivalent diameter is 0.25 inch; and (6) the reactor free flow ratio f is 0.35. The equation (reference 4) for reactor maximum wall temperature based on these assumptions is then

$$T_m = T_w + \frac{1}{2}\left(\frac{Q}{fA_f}\right)_r\left[\left(\frac{1}{\rho V_r c_p}\right)_l + \frac{1}{2h_l\left(\frac{l}{d}\right)_r}\right] \tag{8}$$

where (reference 7)

$$h_l = \frac{k_l}{d_r}\left[7 + 0.025\left(\frac{\rho V_r c_p d_r}{k}\right)_l^{0.8}\right] \tag{9}$$

The reactor maximum wall temperature corresponding to gross weights optimized for a range of heat-exchanger effective wall temperatures was then determined from equation (8).

RESULTS AND DISCUSSION

The performance of the liquid-metal nuclear-powered turbine-propeller cycle is presented by discussing the effect of design variables on engine performance, and by discussing the performance of the airplane-engine combination in terms of airplane gross weight and reactor heat-release rate and total engine air flow.

Engine Performance

The net thrust per engine plus heat-exchanger weight of the turbine-propeller cycle is optimized for heat-exchanger inlet Mach number, compressor pressure ratio, and turbine-inlet temperature for a range of heat-exchanger effective wall temperatures. Results are presented for an altitude of 30,000 feet and a flight Mach number of 0.5. In addition, net thrust per pound of air per second, and heat addition per second per pound of air per second are shown for the corresponding engine conditions to specify engine performance completely.

Heat-exchanger inlet Mach number. - The effect of heat-exchanger air inlet Mach number on thrust per engine plus heat-exchanger weight is illustrated in figure 3. For each assumed value of heat-exchanger effective wall temperature, turbine-inlet temperature, and compressor pressure ratio there is a value of inlet Mach number which gives maximum F_n/W_T.

The optimum inlet Mach numbers range from about 0.12 to 0.15 for all the compressor pressure ratios, turbine-inlet temperatures, and exchanger wall temperatures considered. At optimum compressor pressure ratio and turbine-inlet temperature, the best inlet Mach number is very close to 0.12 for the range of exchanger wall temperatures T_W investigated.

The corresponding total-pressure ratio across the heat exchanger at optimum inlet Mach numbers is about 0.92 for the range of exchanger wall temperatures considered.

2480

<u>Optimum compressor pressure ratio</u>. - The net thrust per engine-plus-heat-exchanger weight for various values of exchanger effective wall temperature, turbine-inlet temperature, and for optimum inlet Mach number is shown as a function of compressor pressure ratio in figure 4. The corresponding values of thrust per pound of air per second are also shown. The solid lines represent constant turbine-inlet temperature and the dashed lines are the envelope curves for maximum F_n/W_T at any pressure ratio. The figures indicate that the optimum compressor pressure ratio is about 4 for an effective wall temperature of 1200° R and increases to 12 for a wall temperature of 2000° R for operation at optimum turbine-inlet temperature.

<u>Optimum turbine-inlet temperature</u>. - For each assumed heat-exchanger effective wall temperature T_w and compressor pressure ratio P_2/P_1, there is a value of turbine-inlet temperature T_3 which gives maximum F_n/W_T. As observed in figures 4(a) to 4(e), the optimum T_3 is very close to T_w.

<u>Heat addition per second per pound of air per second</u>. - The heat addition per second per pound of air flow per second through the engine is shown in figure 5 as a function of compressor pressure ratio for a range of turbine-inlet temperatures. This figure is included to enable the calculation of reactor heat-release rates from equation (7).

<u>Effect of heat-exchanger effective wall temperature on optimum engine performance</u>. - The optimum performance of the engine is summarized in figure 6. Maximum thrust per engine-plus-heat-exchanger weight F_n/W_T is plotted against T_w. The corresponding values of F_n/w_a, Q/w_a, optimum compressor pressure ratio $(P_2/P_1)_{opt}$, optimum turbine-inlet temperature $(T_3)_{opt}$, and the difference between the heat-exchanger effective wall temperature and optimum turbine-inlet temperature $T_w-(T_3)_{opt}$ are also shown as a function of T_w.

As previously shown, optimum P_2/P_1 varies from 4 to 12 for the range of T_w given.

The temperature difference $T_w-(T_3)_{opt}$ varies from 48° to 87° R over the range of T_w given. This temperature difference is small in the case of the turbine-propeller cycle because the exchanger weight is relatively small compared with the compressor, turbine, gearbox, and propeller weights and, consequently, a longer exchanger is desirable to obtain higher turbine-inlet temperatures up to the limit where the resulting increase in pressure drop and weight causes the thrust per engine plus exchanger weight to decrease.

Airplane Performance

The airplane performance in terms of minimum gross weight and reactor heat release was calculated from equations (6) and (7) using the maximum values of F_n/W_T for several values of heat-exchanger effective wall temperatures. Inasmuch as gross weight is minimum for maximum F_n/W_T, the gross weights shown are minimum for the given values of heat-exchanger effective wall temperature. In the case of the turbine-propeller cycle, the difference between the reactor maximum wall temperature and heat-exchanger effective wall temperature is small so that the minimum gross weight calculated for each exchanger wall temperature is close to the minimum gross weight for the corresponding reactor maximum wall temperature.

2480

Minimum reactor heat release occurs at somewhat different values of P_2/P_1 and T_3 than does minimum gross weight. The difference in gross weight and reactor heat release for these two cases is slight, however, and therefore no attempt was made to show airplane gross weight for minimum reactor heat release rates.

Effect of heat-exchanger effective wall temperature on airplane gross weight. - The effect of heat-exchanger effective wall temperature on airplane gross weight factor for optimum engine operating conditions is shown in figure 7. For a given value of W_K, the gross weight increases at an increasing rate as the wall temperature is reduced. This rate of increase becomes quite large when a wall temperature of 1200° R is reached. For example, if W_K is selected to be 190,000 pounds, the gross weight varies from 320,000 pounds at a T_W of 2000° R to 367,000 pounds at 1200° R. Decreasing W_K to 150,000 pounds at a T_W of 1200° R reduces the gross weight from 367,000 pounds to 290,000 pounds.

Figure 8 shows the ratio of reactor heat release to W_K plotted against heat-exchanger effective wall temperature. The reactor heat release increases at an increasing rate with a decrease in wall temperature. This rate of increase becomes quite large for wall temperatures below 1200° R. This is to be expected inasmuch as the gross weight increases quite rapidly in the same range. For a value of W_K of 190,000 pounds, the heat release varies from 33,300 Btu per second at a T_W of 2000° R to 74,500 Btu per second at 1200° R. Decreasing W_K to 150,000 pounds at a T of 1200° R reduces the heat release rate from 74,500 to 59,000 Btu per second.

Effect of heat-exchanger effective wall temperature on total engine air flow and frontal area. - The ratio of total engine air flow to W_K is shown in figure 9 as a function of heat-exchanger effective wall temperature. The air flow increases rapidly with decreasing T_W. For example, the total air flow at 1200° R is five times the value at 2000° R.

Figure 10 was prepared to show the ratio of frontal area to W_K of the compressor, heat exchanger, turbine outlet, and exhaust-nozzle outlet as functions of heat-exchanger effective wall temperature for optimum engine operation. The turbine-outlet area is largest over the full range of temperature shown. The heat-exchanger frontal area is very low at the high temperature because of the high optimum pressure ratio P_2/P_1. As the temperature is reduced, the heat-exchanger frontal area increases very rapidly because of the reduction of optimum P_2/P_1 and approaches the turbine exit area at a T_w of 1200° R.

2480

Effect of W_K and heat-exchanger effective wall temperature on reactor maximum wall temperature. - The reactor maximum wall temperature T_m is shown as a function of W_K in figure 11 for a range of heat-exchanger effective wall temperatures, and for three reactor-to-air heat-exchanger systems. This curve relates T_m to T_w so that given T_w and the optimum airplane performance as determined from figures 7 to 10, the corresponding T_m can be obtained.

The solid lines represent the binary lithium-air system, which has been generally considered in this report, and show the value of T_m required for a given T_w as W_K (and consequently Q) increases.

The dashed lines represent two intermediate heat-exchanger systems described in the appendix, and show the penalties in required T_m associated with these systems. It can be seen that in the range of W_K of current interest (100,000 to 150,000 lb) that this penalty is small particularly at the higher values of T_w.

Effect of reactor maximum wall temperature and W_K on airplane gross weight and reactor heat release rate. - Airplane gross weight is shown as a function of reactor heat release rate in figure 12 for a range of reactor maximum wall temperatures and a range of W_K for the binary lithium-air system. This plot summarizes the airplane performance at optimum engine operating conditions. For any values of W_K and T_m, the gross weight and heat release are found directly.

Detailed performance and weight breakdown at several operating temperatures for turbine-propeller airplane. - Table I shows airplane and engine performance, giving a more detailed listing of component weights and operating conditions than is shown elsewhere in the present report. The table is presented for the flight condition considered previously and for several combinations of T_w and T_3.

SUMMARY OF RESULTS

An analysis of the nuclear-powered liquid-metal turbine-propeller cycle is presented for a range of engine operating conditions, an altitude of 30,000 feet, and a flight Mach number of 0.5. The following results were obtained from the investigation:

2480

1. For optimum compressor pressure ratio and turbine-inlet temperature, the optimum heat-exchanger inlet Mach number is very close to 0.12 for heat-exchanger effective wall temperatures of 1200° to 2000° R.

2. The optimum compressor pressure ratio at optimum turbine-inlet temperature varies from about 4 at 1200° R heat-exchanger effective wall temperature to 12 at 2000° R.

3. The difference between the heat-exchanger effective wall temperature and the optimum turbine-inlet temperature varies from 48° R at a heat-exchanger effective wall temperature of 1200° R to 84° R at 2000° R.

4. Airplane gross weight and reactor heat release rates at the flight conditions considered are quite insensitive to heat exchanger effective wall temperatures as low as 1200° R. For a value of reactor plus shield plus auxiliary equipment weight of 190,000 pounds, the airplane gross weights and reactor heat release rates required by this cycle ranged from about 360,000 pounds and 74,500 Btu per second at a wall temperature of 1200° R T_w to 320,000 pounds and 33,300 Btu per second at 2000° R. For lower values of the sum of reactor plus shield plus auxiliary weight, the gross weight and reactor heat release rate decrease proportionately.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, February, 7, 1952

APPENDIX - REACTOR MAXIMUM WALL TEMPERATURE FOR TERNARY CYCLE

Two ternary cycles (lithium-lithium-air and lithium-sodium hydroxide-air) were considered to show their effect on reactor maximum wall temperature. No attempt was made to optimize the intermediate heat-exchanger configuration which was assumed to be the same for both cycles, however, representative increases in reactor wall temperature were obtained.

2480

The heat exchanger was considered to consist of steel tubes 0.25 inch in diameter with 0.01-inch wall thickness. The tubes were spaced so that the flow area between the tubes was equal to the flow area in the tubes. The total flow area in the exchanger tubes was assumed equal to that of the reactor where the reactor had a free flow factor of 0.35, and a diameter and length of 2.5 feet.

Lithium-lithium-air cycle. - The velocity of the lithium in the reactor and on both sides of the exchanger for this case was assumed to be 15 feet per second. Consequently,

$$(\rho V_x A c_p)_{\text{primary lithium}} = (\rho V_x A c_p)_{\text{secondary lithium}} \tag{A1}$$

and the temperature difference between the primary and secondary lithium is constant along the exchanger length. This constant ΔT is given by:

$$\Delta T = \frac{Q}{US} \tag{A2}$$

where

$$\frac{1}{US} = \frac{1}{(hS)_{\text{Li primary}}} + \frac{1}{(hS)_{\text{Li secondary}}} + \frac{X}{(kS_{av})_{\text{tube wall}}} \tag{A3}$$

X is tube-wall thickness, and h is determined from equation (9) in the text.

The new reactor wall temperature is found by adding the ΔT (equation (A2)) to T_m found by equation (8) in the text.

Sodium hydroxide - lithium-air cycle. - The free flow factor for this reactor was assumed to be 0.70. The sodium hydroxide velocity was maintained at 15 feet per second in the reactor; consequently, its velocity in the intermediate heat-exchanger tubes is 30 feet per second. The secondary coolant (lithium) velocity is maintain at 15 feet per second. In this case

$$(\rho V_x A c_p)_{\text{primary NaOH}} \neq (\rho V_x A c_p)_{\text{secondary lithium}} \tag{A4}$$

Therefore the temperature difference between the primary and secondary coolant was not constant along the exchanger length. The log mean temperature difference is given by:

$$\Delta T = \frac{Q}{US} = \frac{\left(T_{NaOH_{in}} - T_{Li_{out}}\right) - \left(T_{NaOH_{out}} - T_{Li_{in}}\right)}{\ln \dfrac{\left(T_{NaOH_{in}} - T_{Li_{out}}\right)}{\left(T_{NaOH_{out}} - T_{Li_{in}}\right)}} \tag{A5}$$

2480

where

$$\frac{1}{US} = \frac{1}{(hS)_{NaOH\ primary}} + \frac{1}{(hS)_{Li\ secondary}} + \left(\frac{X}{kS_{av}}\right)_{tube\ wall} \tag{A6}$$

and X is tube-wall thickness.

The Prandtl number for NaOH is close to 1 and therefore, the following equation for h was used (reference 8).

$$h_{NaOH} = 0.023\ \frac{k}{d_t}\ (Re)^{0.8}\ (Pr)^{0.4} \tag{A7}$$

The last two terms in equation (A6) are the same as the last two terms in equation (A3). The new reactor wall temperature is found by adding the ΔT (equation (A5)) to T_m found by equation (8) in the text where the liquid coolant properties are those of NaOH, the reactor free flow factor f is 0.70, and the coolant flow velocity in the reactor V_r is 15 feet per second.

REFERENCES

1. Doyle, Ronald B.: Calculated Performance of Nuclear Turbojet Powered Airplane at Flight Mach Number of 0.9. NACA RM E50B23, 1950.

2. Humble, L. V., Wachtl, W. W., and Doyle, R. B.: Preliminary Analysis of Three Cycles for Nuclear Propulsion of Aircraft. NACA RM E50H24, 1950.

3. Doyle, R. B.: Calculated Performance of a Direct-Air Nuclear Turbojet-Powered Airplane using a Split-Flow Reactor and a Separated-Type Shield. NACA RM E50K06, 1950.

4. Wachtl, William W., and Rom, Frank E.: Analysis of the Liquid-Metal Turbojet Cycle for Propulsion of Nuclear Powered Aircraft. NACA RM E51D30, 1951.

5. Pinkel, Benjamin, Noyes, Robert N., and Valerino, Michael F.: Method for Determining Pressure Drop of Air Flowing Through Constant-Area Passages for Arbitrary Heat-Input Distributions. NACA TN 2186, 1950.

6. Trout, Arthur M., and Hall, Eldon W.: Methods for Determining Optimum Division of Power Between Jet and Propeller for Maximum Thrust and Power of a Turbine-Propeller Engine. NACA TN 2178, 1950.

7. Lyon, Richard N.: Forced Convection Heat Transfer Theory and Experiments with Liquid Metals. ORNL, Aug. 1949.

8. McAdams, William H.: Heat Transmission. McGraw-Hill Book Co., Inc., 2d ed., 1942.

2480

TABLE I - WEIGHT AND PERFORMANCE BREAKDOWN FOR BINARY CYCLE

Flight condition				
Altitude, ft	30,000	30,000	30,000	30,000
Flight Mach number	0.5	0.5	0.5	0.5
Design point, L/D	20	20	20	20
Operating point				
Heat exchanger wall temperature, T_w, ^{o}R	1,200	1,400	1,600	2,000
Optimum turbine-inlet temperature, T_3, ^{o}R	1,152	1,342	1,523	1,913
Optimum compressor pressure ratio, P_2/P_1	4.2	6.25	8.30	12.4
Component weights, lb				
Reactor + shield + payload + auxiliary	190,000	190,000	190,000	190,000
Airplane structure	128,800	119,300	115,000	111,700
Engines	25,500	16,100	11,400	6,700
Heat exchangers	14,400	6,300	3,750	1,500
Propeller + gear box	10,500	10,000	9,850	9,600
Airplane gross weight	369,200	341,700	330,000	319,500
Reactor				
Diameter equals length, ft	2.5	2.5	2.5	2.5
Heat release, Btu/sec	74,500	51,300	41,750	33,300
Maximum wall temperature, T_m, ^{o}R	1,237	1,425	1,621	2,017
Lithium temperature into reactor, ^{o}R	1,166	1,376	1,581	1,985
Lithium temperature out of reactor, ^{o}R	1,234	1,422	1,619	2,015
Lithium velocity through reactor, ft/sec	15	15	15	15
Engine				
Net thrust per engine plus exchanger weight, lb/lb	0.375	0.535	0.675	0.90
Total net thrust per air flow, lb/(lb air)(sec)	28.6	46.5	68.0	119.3
Total engine air flow, lb/sec	643	367	243	134
Total net thrust, lb	18,400	17,050	16,500	15,980
Engine jet thrust, lb	11,750	7,040	4,810	2,950
Propeller thrust, lb	16,600	15,690	15,450	15,100
Propeller shaft horsepower, HP	16,700	15,780	15,550	15,200
Compressor frontal area, ft^2	68	38.8	25.7	14.1
Turbine exit area, ft^2	80.5	47.6	32.4	19.25
Exhaust-nozzle outlet area, ft^2	71.2	42.0	28.7	17.0
Heat exchanger				
Lithium temperature in, ^{o}R	1,234	1,422	1,619	2,015
Lithium temperature out, ^{o}R	1,166	1,376	1,581	1,985
Air inlet Mach number	0.12	0.12	0.12	0.12
Air inlet temperature, T_2, ^{o}R	687.5	780.3	853	966
Air outlet temperature, T_3, ^{o}R	1,152	1,342	1,533	1,913
Heat addition to air per pound, Q/w_a, Btu/lb air	115	140	172	248
Core frontal area, ft^2	74.4	30.4	15.8	6.2
Air total-pressure ratio across exchanger, P_3/P_2	0.932	0.930	0.925	0.916

2480

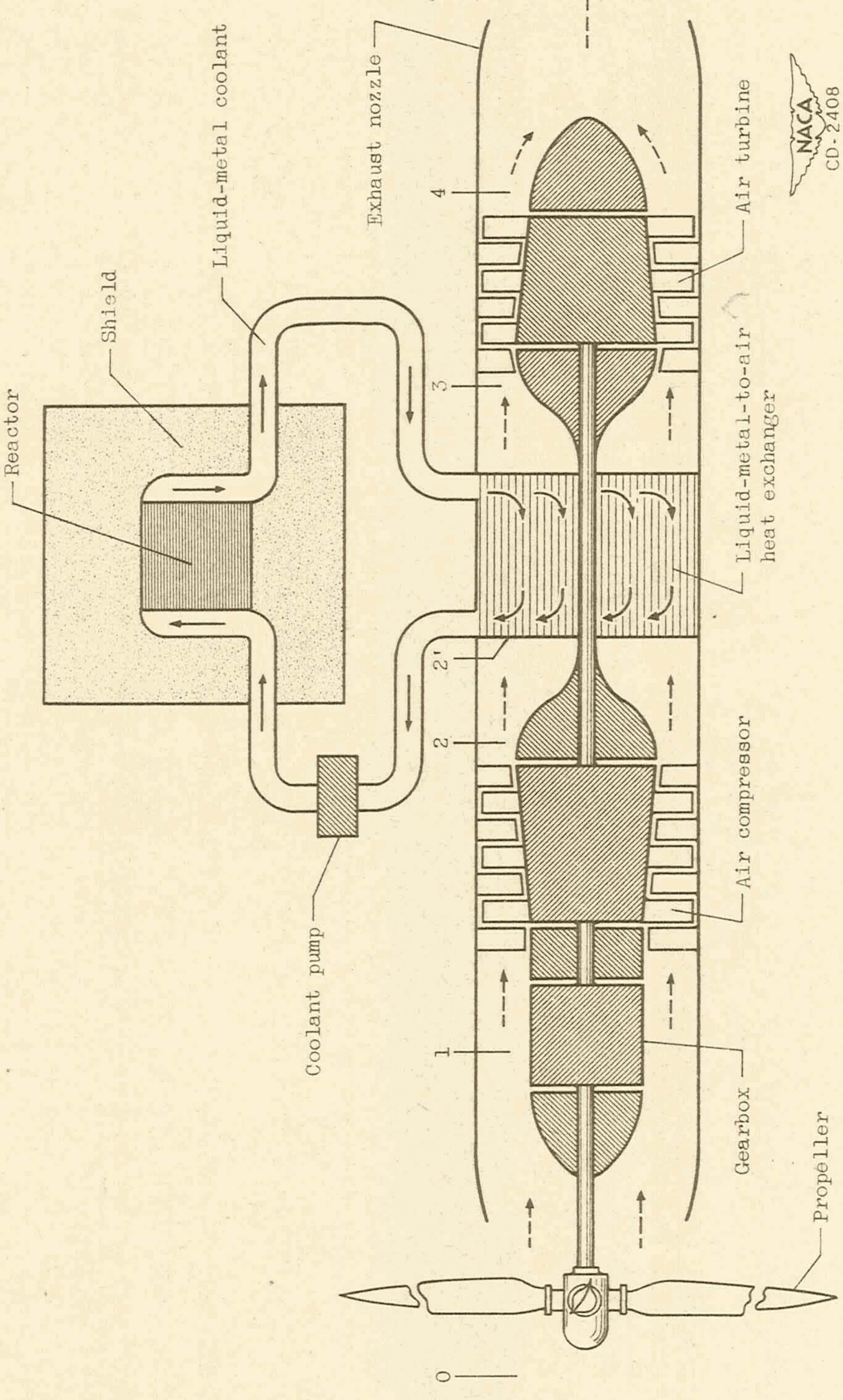


Figure 1. - Schematic diagram of liquid-metal nuclear-powered-turbine-propeller engine.

2480

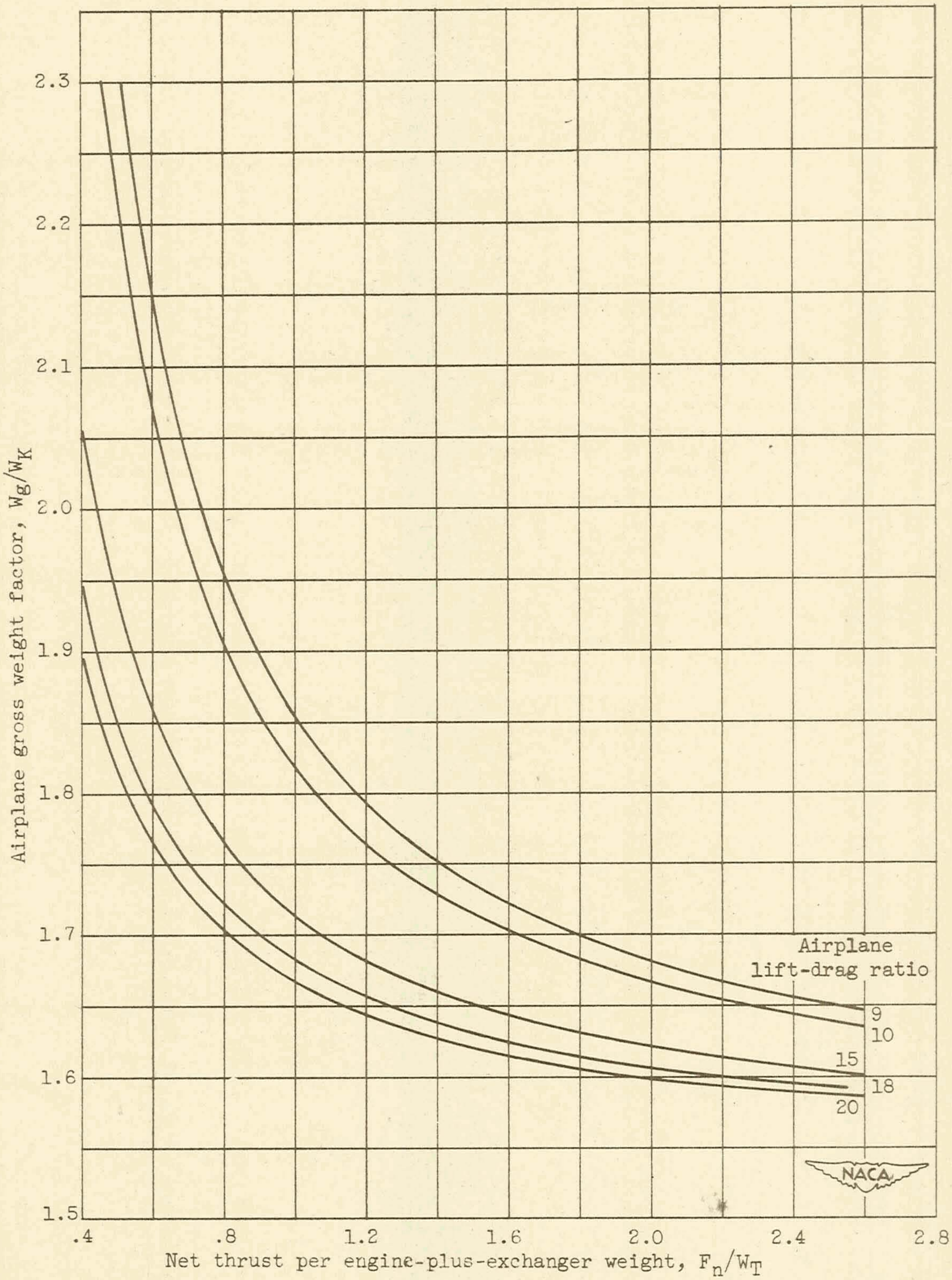


Figure 2. - Airplane gross weight factor as function of net thrust per engine-plus-exchanger weight and airplane lift-drag ratio.

2480

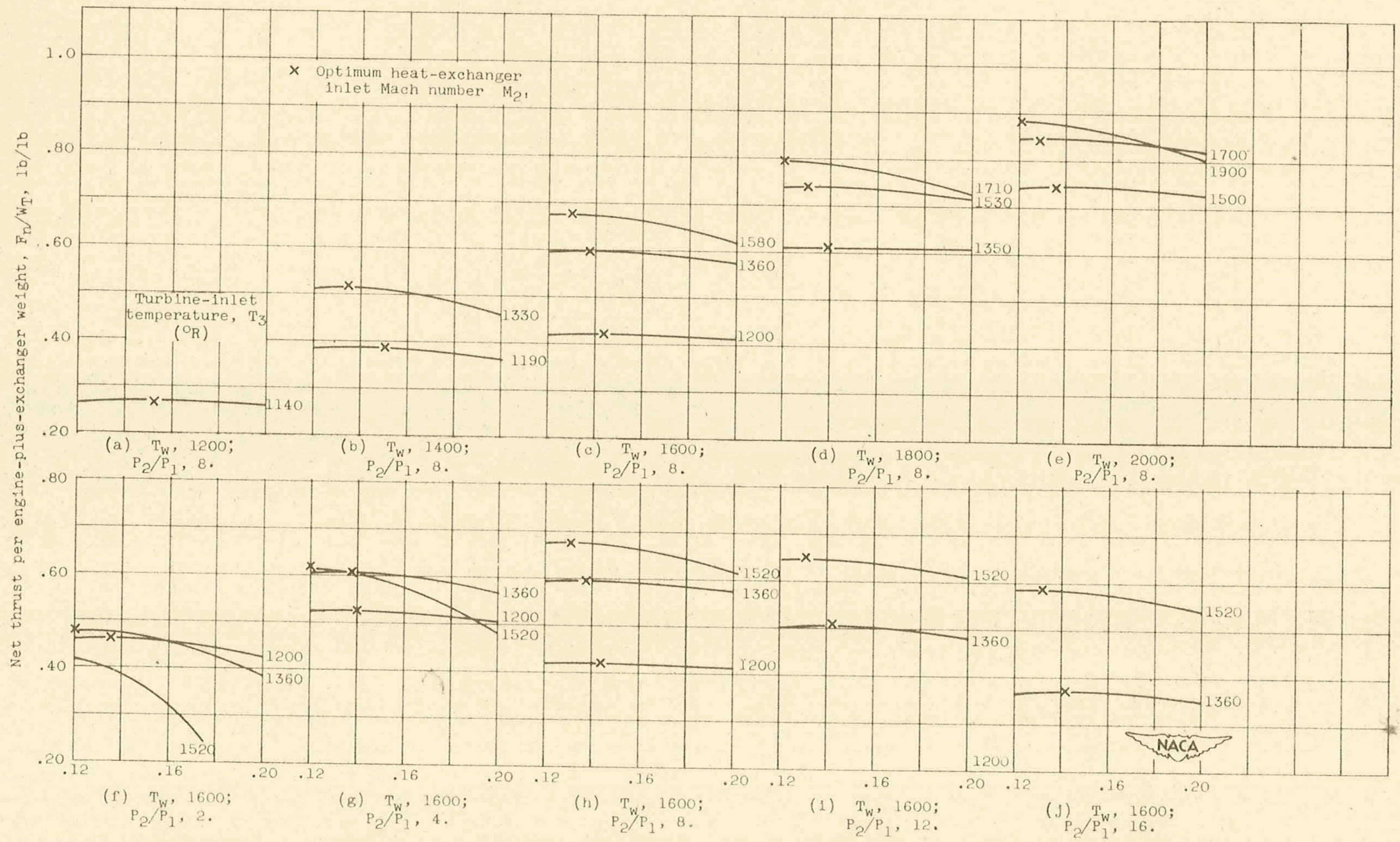
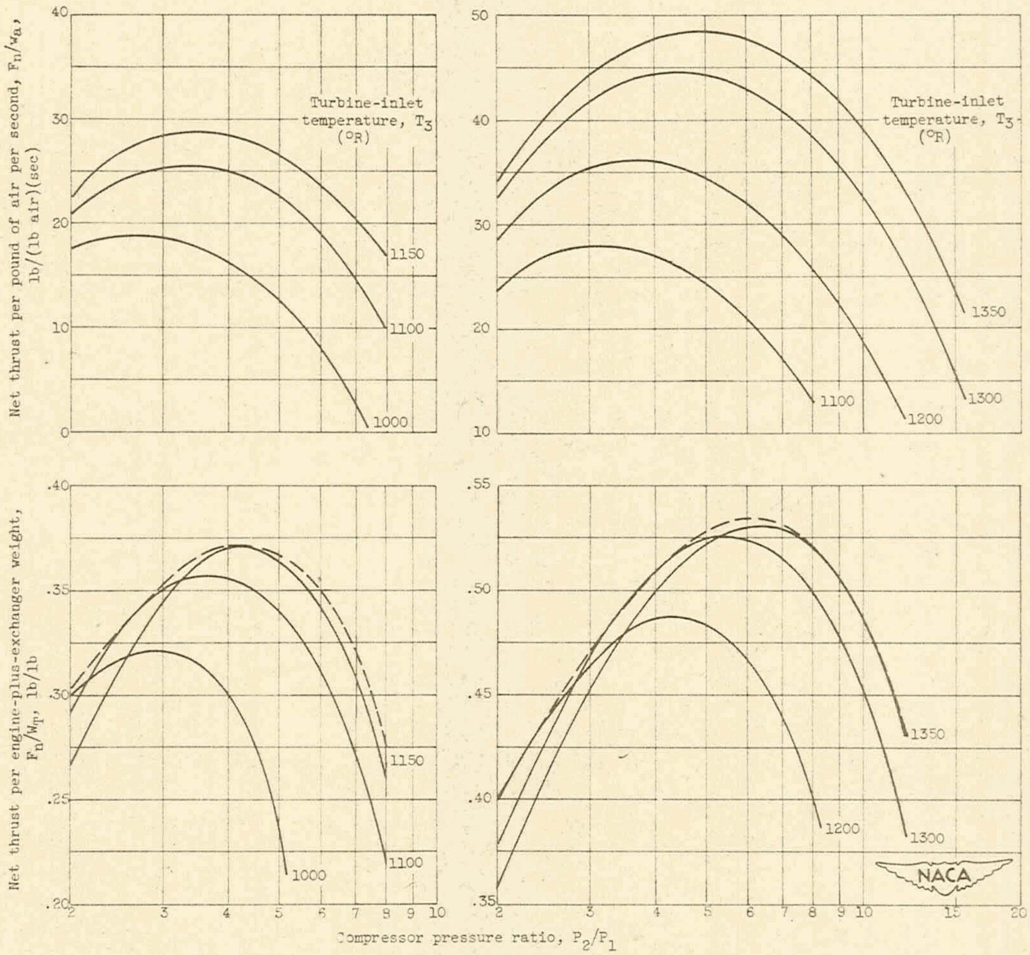


Figure 3. - Net thrust per engine-plus-heat-exchanger weight as function of heat-exchanger inlet Mach number for various values of heat-exchanger effective wall temperature; turbine-inlet temperature, and compressor pressure ratio. Altitude 30,000 feet; flight Mach number, 0.5.

2480

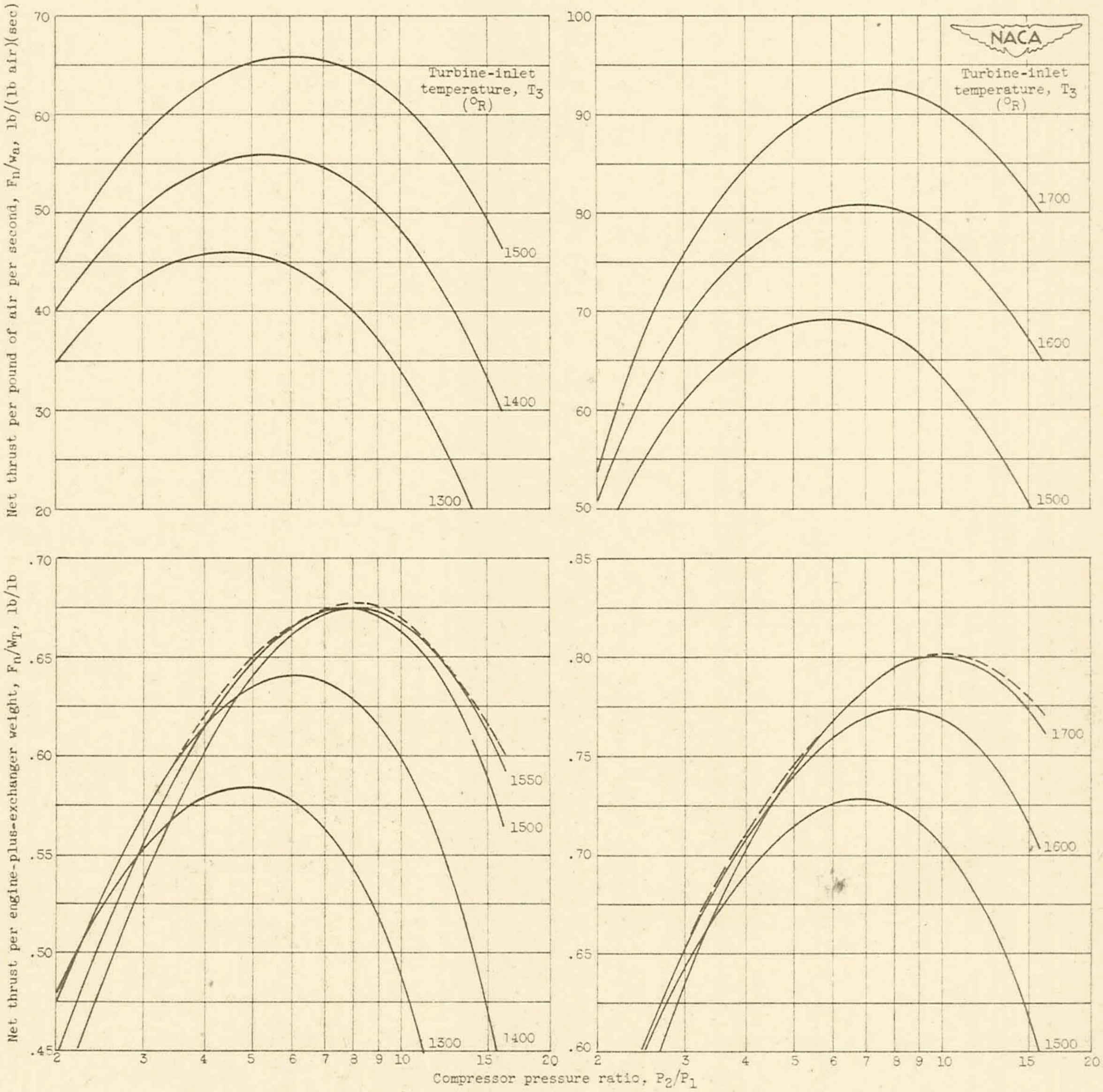


(a) Heat exchanger effective wall temperature, 1200° R

(b) Heat exchanger effective wall temperature, 1400° R.

Figure 4. - Net thrust per engine-plus-heat-exchanger weight and net thrust per pound of air per second as functions of compressor pressure ratio, heat-exchanger effective wall temperature, and turbine-inlet temperature. Optimum heat-exchanger inlet Mach number; altitude, 30,000 feet; flight Mach number, 0.5. Dashed envelop curve indicates maximum net thrust per engine-plus-exchanger weight at any compressor pressure ratio.

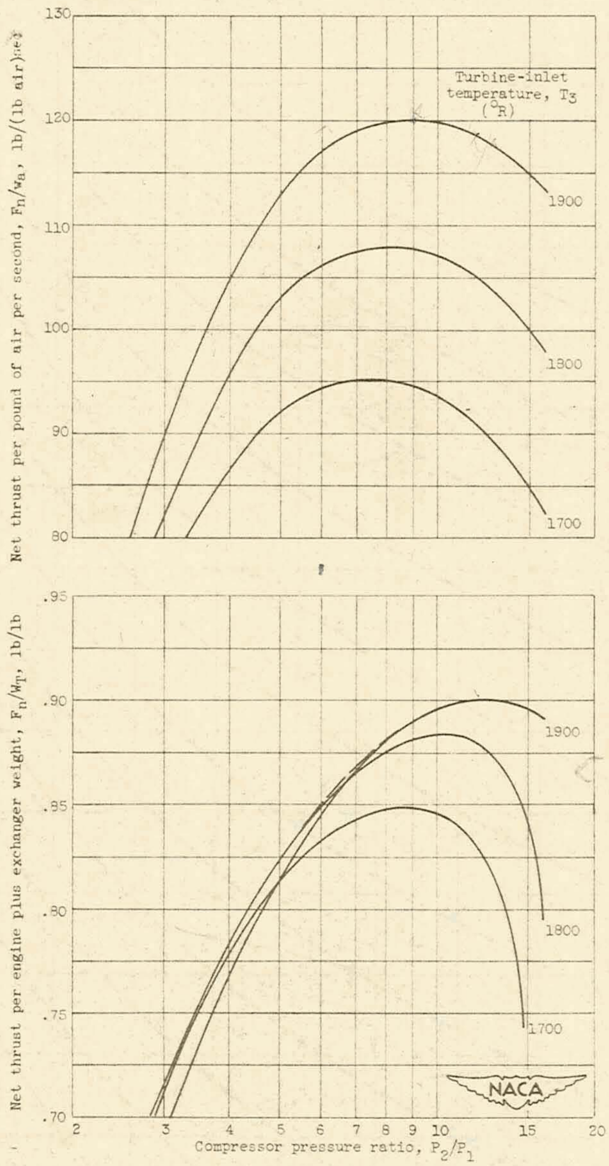
2480



(c) Heat exchanger effective wall temperature, 1600° R.

(d) Heat exchanger effective wall temperature, 1800° R.

Figure 4. - Continued. Net thrust per engine-plus-heat-exchanger weight and net thrust per pound of air per second as functions of compressor pressure ratio, heat-exchanger effective wall temperature, and turbine-inlet temperature. Optimum heat-exchanger inlet Mach number; altitude, 30,000 feet; flight Mach number, 0.5. Dashed envelop curve indicates maximum net thrust per engine-plus-exchanger weight at any compressor pressure ratio.



(e) Heat-exchanger effective wall temperature, 2000° R.

Figure 4. - Concluded. Net thrust per engine-plus-heat-exchanger weight and net thrust per pound of air per second as functions of compressor pressure ratio, heat-exchanger effective wall temperature, and turbine-inlet temperature. Optimum heat-exchanger inlet Mach number; altitude, 30,000 feet; flight Mach number, 0.5. Dashed envelop curve indicates maximum net thrust per engine-plus-exchanger weight at any compressor pressure ratio.

2480

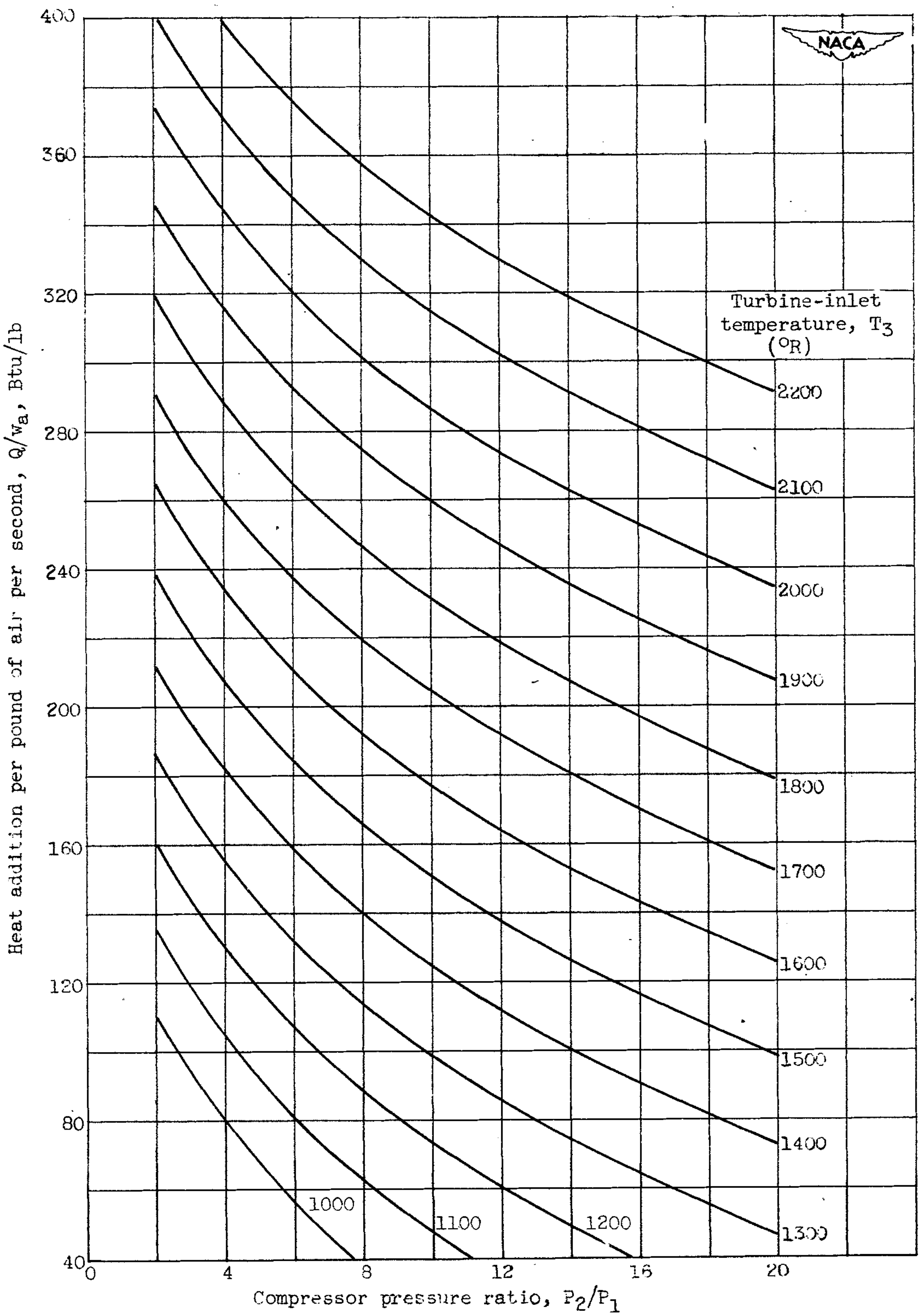


Figure 5. - Heat addition per second per pound of air per second as function of compressor pressure ratio, and turbine-inlet temperature. Altitude 30,000 feet; flight Mach number, 0.5.

[redacted]

2480

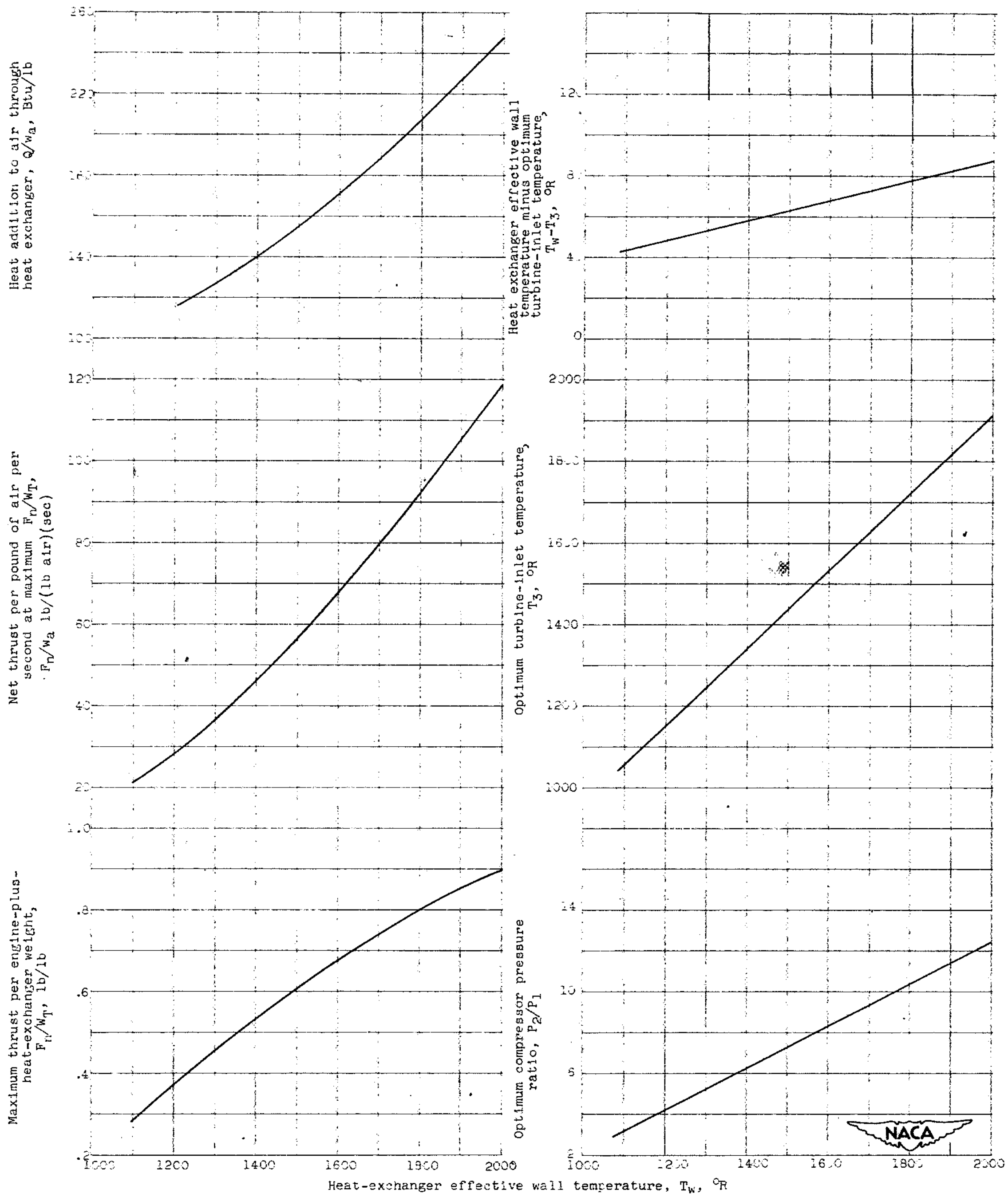


Figure 6. - Optimum engine performance as a function of heat-exchanger effective wall temperature. Optimum heat-exchanger inlet Mach number; altitude 30,000 feet; flight Mach number, 0.5.

[redacted]

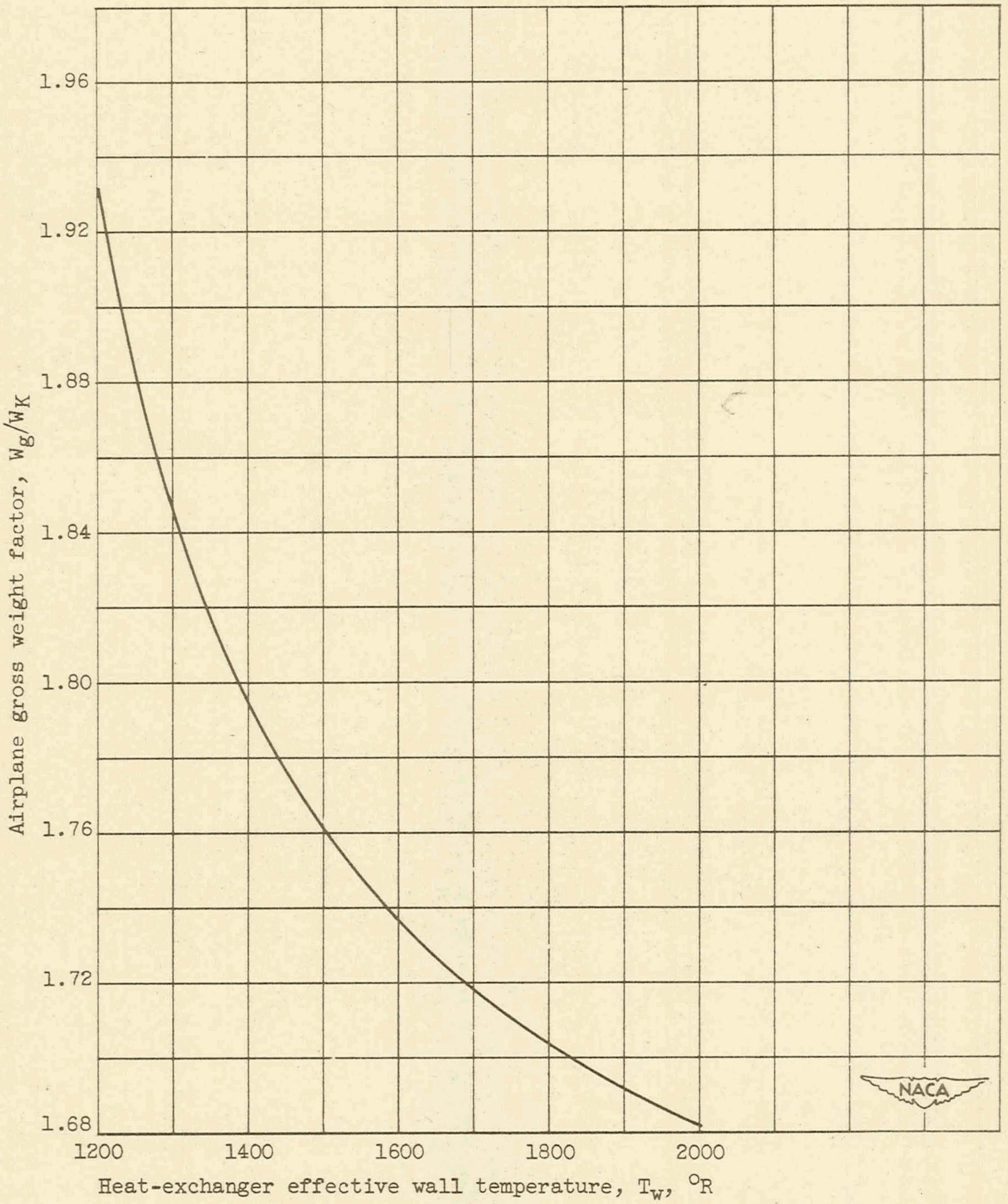


Figure 7. - Airplane gross weight factor as function of heat-exchanger effective wall temperature. Optimum compressor pressure ratio; optimum heat-exchanger inlet Mach number; optimum turbine-inlet temperature; ratio of structure weight to gross weight, 0.35; altitude, 30,000 feet; flight Mach number, 0.5.

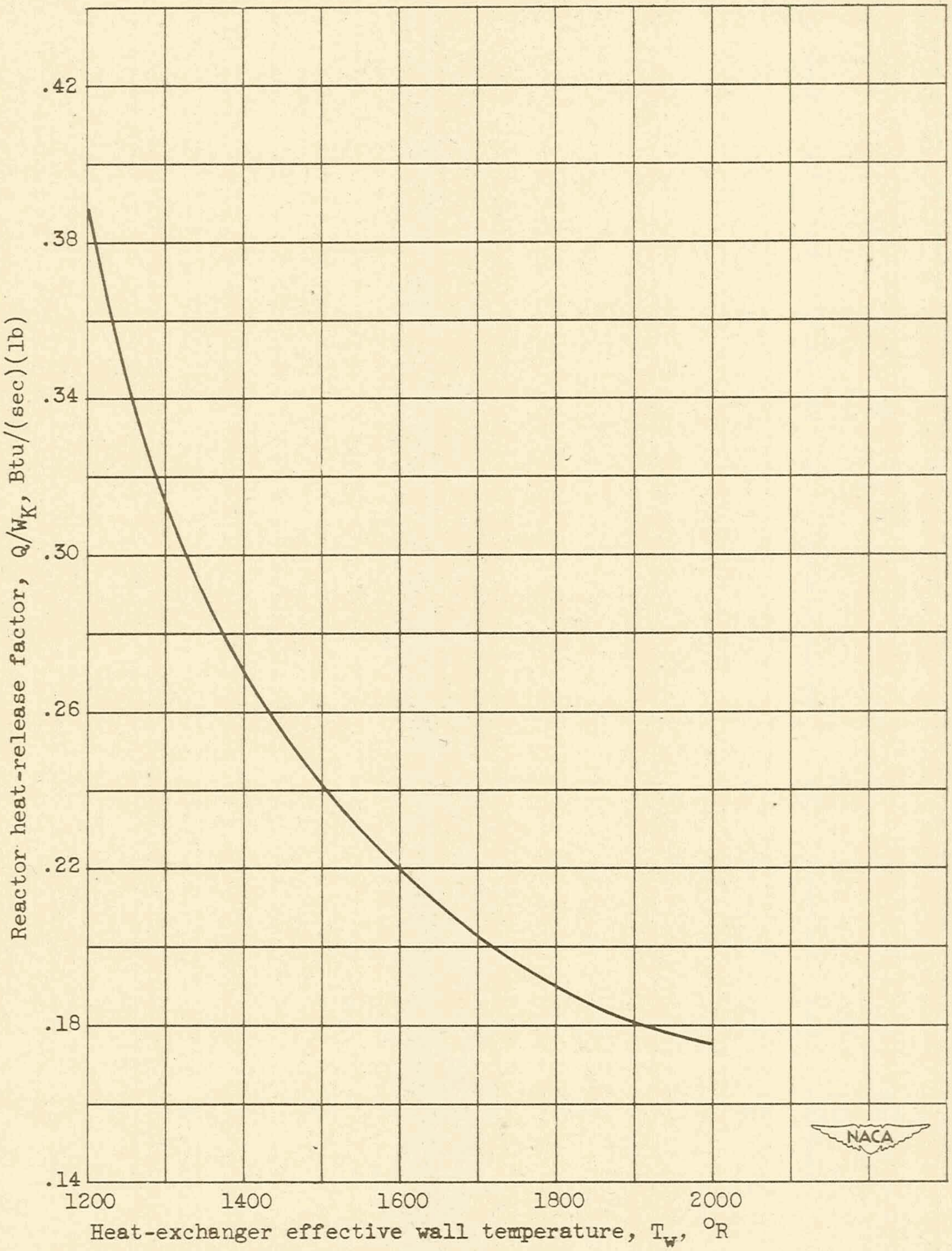


Figure 8. - Reactor heat-release factor as function of heat-exchanger effective wall temperature. Optimum compressor pressure ratio; optimum turbine-inlet temperature; ratio of structural weight to gross weight, 0.35; altitude, 30,000 feet; flight Mach number, 0.5.

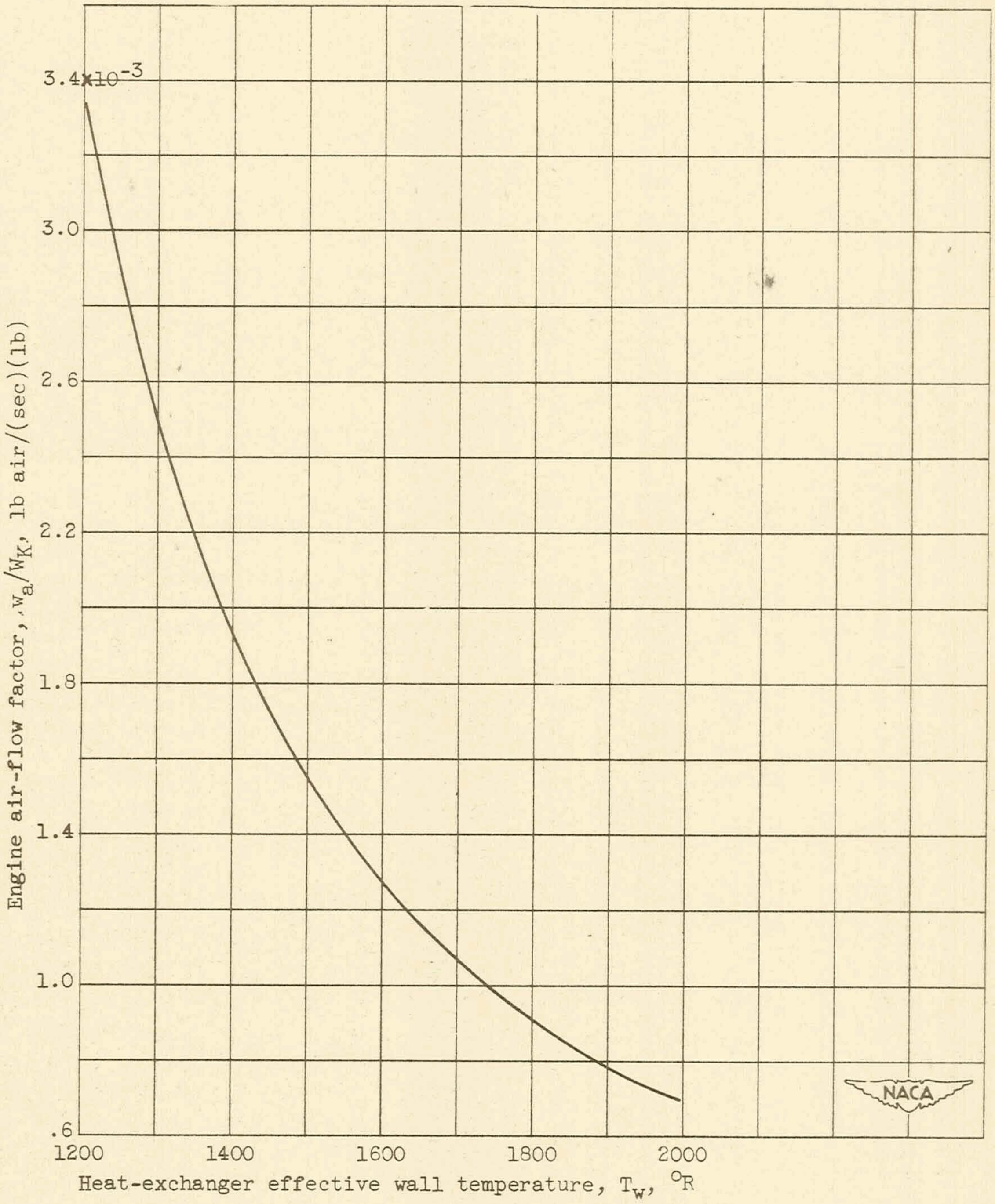


Figure 9. - Engine air-flow factor as function of heat-exchanger effective wall temperature. Optimum heat-exchanger inlet Mach number; optimum turbine-inlet temperature; ratio of structural weight to gross weight, 0.35; altitude, 30,000 feet; flight Mach number, 0.5.

2480

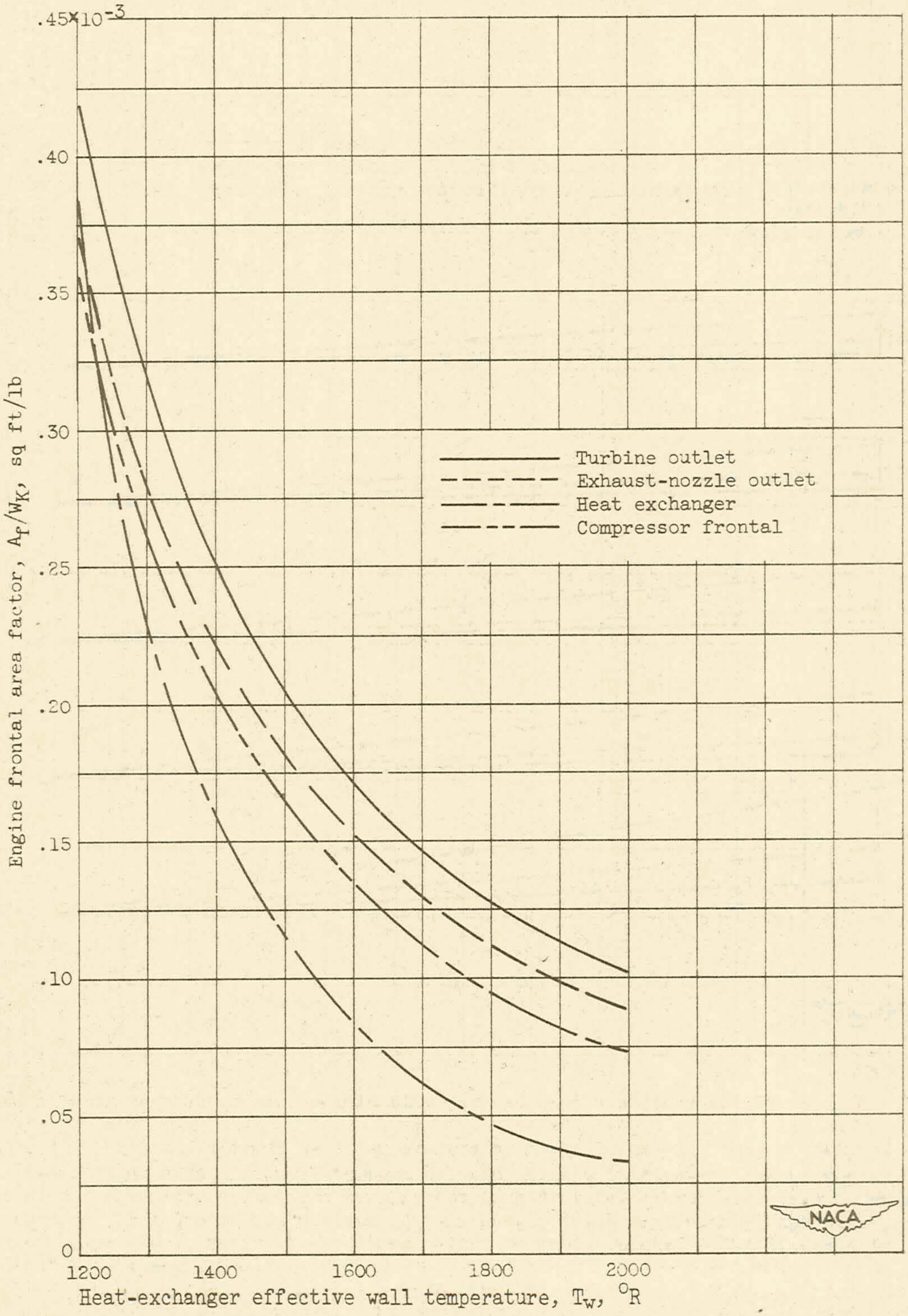


Figure 10. - Engine frontal area factor as function of heat-exchanger effective wall temperature. Optimum compressor pressure ratio; optimum heat-exchanger inlet Mach number; optimum turbine-inlet temperature; ratio of structure weight to gross weight, 0.35; altitude, 30,000 feet; flight Mach number, 0.5.

[redacted]

[redacted]

[redacted]

[redacted]